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NACA RM L50L06

RESEARCH MEMORANDUM

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FREE-JET TESTS OF A 6.5-INCH-DIAMETER RAM-JET ENGINE AT MACH NUMBERS OF 1.81 AND 2.00

By Maxime A. Faget, Raymond S. Watson, and Walter A. Bartlett, Jr.

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Langley Field, Va.

Classification cancelled (or changed to Unclassified)

By authority Nasa Tech Pub Announcement #101
(OFFICER AUTHORIZED TO CHANGE)

By .... 25 May 56

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

March 7, 1951

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319.98/13

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

FREE-JET TESTS OF A 6.5-INCH-DIAMETER RAM-JET ENGINE AT MACH NUMBERS OF 1.81 and 2.00

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SUMMARY

A 6.5-inch-diameter ram-jet engine, having a design Mach number of 2.13 and a short-flame-length combustor, was tested in free supersonic jets. Performance characteristics of the engine are presented for Mach numbers of 1.81 and 2.00 over a fuel-air-ratio range of 0.015 to 0.100 to supplement data obtained in a flight test of this engine reported in NACA RM L50H10.

A maximum thrust coefficient of 0.925 was obtained at a Mach number of 2.00 for a free-stream temperature of -79° F and a fuel-air ratio of 0.070. This measured thrust coefficient was more than adequate to overcome the drag of the ram-jet flight-test missile at a Mach number of 2.00 (as described in NACA RM L50H10). At this test condition the total-pressure recovery in the diffuser was 84 percent of free-stream total pressure.

The maximum air impulse and combustion efficiencies were 93 percent and 81 percent, respectively, near a fuel-air ratio of 0.030. Tests in which the free-stream temperature was varied from sea level to stratosphere conditions indicated that no appreciable change in combustion efficiency occurred at any given fuel-air ratio. The lowest value of specific fuel consumption was 2.72 obtained at a Mach number of 2.00 for a free-stream temperature of -79° F and a thrust coefficient of 0.560.

A comparison of ground-test and flight-test performance data showed that values of combustion efficiency and specific fuel consumption were of the same order of magnitude and that diffuser total-pressure recovery for a given thrust coefficient and Mach number was in excellent agreement.

INTRODUCTION

A ram-jet configuration in which the power plants were mounted in the tail surfaces was proposed in reference 1. This reference showed that high performance as well as a large useful volume was obtainable by using such a configuration and indicated the need for a short-flame-length combustor in order to make the configuration practical for small aircraft. In reference 2 the initial flight test of a configuration powered by nacelle-type ram jets mounted on the tail was reported.

Prior to flight testing this configuration, the ram-jet power plants were thoroughly tested in the preflight jet facility at the Pilotless Aircraft Research Station at Wallops Island, Va. This equipment consists of an air supply and of suitable nozzles, which have been provided at the flight-test station so that ground testing can assure proper functioning of the engine and engine components under simulated flight conditions.

During the ground-test phase, a lightweight, 20-inch-combustor-length burner was proven practical. Concurrently, the fuel system for the flight model, which consisted of a programing valve for metering fuel from a high-pressure fuel tank to the engines, was proof-tested. At the same time a dependable, quick-starting ignition technique, applicable both to ground and flight tests, was obtained.

A technique for the reduction of data obtained in such a free jet, where the model size is relatively large with respect to the free jet, was developed, and an investigation of the performance of a ram-jet engine similar to the power plants used in the initial flight test was undertaken. This investigation was undertaken to determine whether the engine gave satisfactory performance at Mach numbers of 1.81 and 2.00. These Mach numbers were considered critical in the flight test, since the model was required to accelerate after being boosted to a Mach number of approximately 1.85. The results of these engine performance tests, made at Mach numbers of 1.81 and 2.00, are presented in this paper.

SYMBOLS

C_T thrust coefficient, based on combustion-chamber area

W_a weight flow of air, pounds per second

W_f weight flow of fuel, pounds per second

T	free-stream static temperature, degrees Fahrenheit
M	Mach number
H	total pressure, pounds per square foot absolute
p	static pressure, pounds per square foot absolute
S_f	specific fuel consumption, pounds of fuel per hour per pound of thrust
η_i	air impulse efficiency
η_c	combustion efficiency
f/a	fuel-air ratio, weight rate of fuel flow to weight rate of air flow
A^*	critical area where sonic velocity is obtained when isentropic flow is assumed, square feet
A	area at any station, square feet
R	universal gas constant
X	force measured on thrust stand, pounds
D	drag, pounds
γ	ratio of specific heats of air
S_a	air specific impulse, seconds
ρ	density of air, slugs per cubic foot
V	velocity, feet per second
g	acceleration due to gravity, 32.2 feet per second per second
F	engine thrust, pounds
y	distance from tip of cone to station, inches
r	radius of cone, inches
Δh	change in enthalpy of air in combustion process, British thermal units per pound

h_c lower heating value of ethylene, 20,400 Btu per pound of fuel

θ angle between original direction of flow and shock wave

G actual jet propulsive force, pounds

G' ideal jet propulsive force, pounds

ϕM ratio of jet impulse at any station to the jet impulse at a sonic station

Subscripts:

0 free stream

1 behind oblique shock

2 entrance, 0.085 square foot

3 minimum section of diffuser, 0.0532 square foot

4 rake, 0.151 square foot

5 exit nozzle throat, 0.180 square foot

6 nozzle exit, performance tests, 0.230 square foot

b base of exit plug, tare tests, 0.150 square foot

c surface of cone

i internal

s any point immediately behind oblique shock

t tare

APPARATUS

Preflight Jet

The tests reported herein were conducted in the preflight jet (fig. 1) located at the Pilotless Aircraft Research Station at Wallops Island, Va. This facility was designed to preflight-test research missiles up to a Mach number of 2.25 at sea-level static pressure and over a range of stagnation temperatures up to 600° F. Air for the operation of the preflight jet is stored in two spheres having a total volume of 25,000 cubic feet and at a pressure of 220 psi absolute. The air is dried to a dew point of -60° F as it is pumped into the spheres by a two-stage 600-cubic-feet-per-minute compressor. A hydraulically actuated valve controls air flow and air pressure from the spheres. The air then passes through a heat exchanger, where it is heated sufficiently to compensate for the adiabatic temperature drop through the supersonic nozzles. The air next passes through a 12- by 12-inch supersonic test nozzle and exhausts to atmosphere. The model is mounted on a high-frequency-response, strain-gage thrust stand in the free jet.

Ram-Jet Engine

A sketch of the ram-jet engine tested and a photograph of the model mounted in the preflight jet are presented as figures 2(a) and 2(b), respectively. The engine, geometrically the same as the flight-test engine described in reference 2, is 50.2 inches long and 6.6 inches in diameter. A Mach number 2.13 design, Ferri type (reference 3) of inlet diffuser with a 40° cone was used.

The area ratio of the combined supersonic and subsonic diffuser was 0.369, based on the area at the entrance lip and the combustion-chamber area. The exit-nozzle contraction ratio of the ram-jet engine was 0.783. Two diametrically opposite circular-arc airfoils with a thickness of 1/2 inch and a chord of 3 inches fastened the inner body to the diffuser shell.

The fuel-cooled "donut" burner and fuel-spray assembly (fig. 2(c)) used was geometrically the same as that in the flight engine of reference 2. The burner assembly consists of four doughnut-shaped rings made of $\frac{1}{32}$-inch Inconel sheet and bent to an outside diameter of 4 inches and an inside diameter of $2\frac{1}{2}$ inches. The forward ring was used as a fuel spray, with the other three acting as flame holders. The rings were constructed as a unit and attached at the rear of the main fuel-feed tube. Fuel entered the ring assembly at its downstream end and passed through and cooled this assembly before entering the fuel-spray ring. Fuel was sprayed through 50 No. 43 drill-size holes located on the fuel-spray ring as follows: 24 equally spaced on the outer rim, 18 equally spaced on the inner rim, and 8 holes, 2 on each fuel-supply tube directly behind the fuel-spray ring.

The inner body, the diffuser shell, and the exit nozzle were constructed of mild steel. The burner assembly and combustor shell were constructed of Inconel.

Fuel Supply and Regulation

Technical grade ethylene (C_2H_4) was used as a fuel. Twenty bottles of ethylene were manifolded and the fuel-feed tube led through a heater to vaporize the fuel completely so that proper metering of the ethylene could be obtained in the calibrated fuel-flow venturi. A remote-control valve was used to regulate fuel flow to the model.

TESTS

A thrust-drag balance was used to determine the thrust of the engine exclusive of external drag. Since the engine was large relative to the size of the jet, the external drag would be different from that which would be obtained in flight. However, the external drag - henceforth called the tare drag - would be the same in both the tare tests and performance tests, as the engine mass flow was the same in both cases and was at the maximum value that the inlet geometry would allow at each test Mach number.

Tare Tests

A tail plug (fig. 2(d)) installed in the model for these tare tests was designed from data obtained in reference 4. This tail plug was designed to give choking at the exit so that an accurate determination of internal drag could be made.

An analysis of the method derived for obtaining tare drag by separating internal drag and base drag of the model from the total drag measured is given in appendix A. Drag data were obtained at Mach numbers of 1.81 and 2.00. The tare drags thus obtained were added to the thrust measurements in the performance tests to give the true engine thrust.

Performance Tests

A satisfactory starting technique for ram-jet engines has been devised to allow ignition of the fuel mixture to a free-stream Mach number of at least 2.25. A magnesium disk blocking 69 percent of the combustion-chamber area is attached to the aft flame holder. This disk reduces the air velocity in the combustion chamber to allow proper mixing of fuel and air prior to ignition. Ignition is accomplished by firing two electric squibs. The starting disk then burns away in approximately 0.3 second. The application of this method of ignition to flight testing is described in reference 2.

After the starting disks had burned away, the fuel flow to the engine was varied in every test from lean to rich and covered the ranges of fuel-air ratios presented in this paper.

The performance test at a Mach number of 1.81 was conducted at a free-stream static temperature of 54° F, while the tests at Mach number 2.00 were conducted at free-stream static temperatures of -79° F, 14° F, and 48° F. The aforementioned temperatures cover the range of static temperatures obtained between a winter stratosphere condition and a mean sea-level temperature at the Pilotless Aircraft Research Station.

INSTRUMENTATION AND MEASUREMENTS

Measurements of thrust of the model were obtained with strain-gage beams and recorded on an oscillograph. Measurements of fuel flow, free-stream total pressure, free-stream stagnation temperature and fuel temperature were obtained with electrical pressure pickups and thermocouples and the data recorded on another oscillograph.

A pressure-survey rake (fig. 2(a)), consisting of six total-pressure and one static-pressure tube, was installed at station 4. The total-pressure tubes were equally spaced - three on each side of the static-pressure tube. The rake was mounted in the duct so that pressure measurements were not influenced by the wake from the innerbody support struts.

Wall static taps at stations 4 and 5 also gave measurements of internal pressure in the engine performance tests (fig. 2(a)). A static-pressure tube was installed in the tail plug during the tare-drag runs so that the base pressure of the tail plug could be obtained. These pressures were recorded on optical recording six-cell manometers. The locations of the stations used in analysis of the data are shown in figure 2(d).

Data obtained on the various instruments were time-correlated with a 10-cycle-per-second timer. The instruments used were accurate to 1 percent of their full-scale range.

RESULTS AND DISCUSSION

The force measured with the thrust stand during drag tests is made up of external drag, internal drag, and base drag. A method of separating the external or tare drag from the total drag measured is presented in appendix A. The tare drag thus obtained has been added to all thrust measurements for the computation of thrust coefficient C_T presented in this paper.

The weight flow of air W_a through this type of ram-jet engine can be calculated by the method presented in appendix B. An experimental verification of this method is also introduced in this appendix.

Thrust coefficients C_T as a function of fuel-air ratio are presented in figure 3 for the test Mach numbers of 1.81 and 2.00. It is noted that, as T_0 is reduced in the $M = 2.00$ tests, higher values of C_T are obtained for a given fuel-air ratio. The thrust coefficients are not the highest that could be obtained if the fuel-air ratios were increased beyond those presented; however, the range of fuel-air ratios tested is considered to cover the useful operating range of the engine. Fuel supply pressure in the various tests limited the range of fuel-air ratios covered. Thrust coefficients of 0.700 and 0.768 were obtained at Mach numbers of 1.81 and 2.00, respectively, at air temperatures corresponding to sea-level conditions. At Mach number 2.00, a thrust coefficient of 0.925 was obtained with the air temperature corresponding to temperatures of the stratosphere. Calculations indicated that these thrust coefficients were more than adequate to accelerate the flight missile. Results of the flight test (reference 2) demonstrated that thrust coefficients required to compensate for the missile drag were 0.47 at $M = 1.81$ and 0.424 at $M = 2.00$.

Thrust coefficients calculated from equations given in appendix C for fuel-air ratios of 0.040 and 0.045 at $M = 2.00$, and for 80-percent combustion efficiency are compared with experimental results in figure 4. The predicted increase in thrust coefficient with decreasing free-stream temperature is substantiated by the experimental data.

The ratio of static pressure p_5 at the throat of the nozzles to free-stream total pressure H_0 is plotted as a function of C_T in figure 5 for the two test Mach numbers. The data show that the ratio p_5/H_0 varies linearly with C_T over the C_T range covered by these tests, irrespective of test static temperatures. This indicates that, over the test range of fuel-air ratios, changes in γ have only little effect on the static-pressure ratio in the nozzle exit. Higher values of p_5/H_0 were obtained at $M = 1.81$ than those of $M = 2.00$.

Integrated averages of faired curves of H_4/H_0 weighted against area are presented in figure 6 as a function of C_T for the two test values of Mach number at the several free-stream static temperatures T_0. The data show that the ratio H_4/H_0 varies linearly with C_T over the C_T range of 0.30 to 0.93. It is noted that, as the ratio p_5/H_0 varied linearly with C_T, the plot of H_4/H_0 against C_T could also be expected to be linear, as was indicated by engine performance calculations. In tests at $M = 2.00$, it can be seen that variation in T_0 between -79° F and 48° F does not affect the relationship between H_4/H_0 and C_T. From flight-test data (reference 2), values of $H_4/H_0 = 0.725$ and $C_T = 0.666$ were shown to have occurred at $M = 2.00$. This point, which is included in figure 6, is in good agreement with the faired curves of ground-test results at $M = 2.00$. Values of H_4/H_0 obtained for $M = 1.81$ are higher than those at $M = 2.00$ at the same value of C_T.

The radial distribution of M_4, at several values of H_4/H_0, is presented in figure 7. It is evident that the Mach number profile becomes more uniform as H_4/H_0 increases, resulting from an increase in back pressure.

The specific fuel consumption S_f, or economy, of the ram-jet engine is illustrated in figure 8 as a function of C_T for the two test values of Mach number. The minimum values of S_f at the two test Mach numbers and the thrust coefficients at which they occurred are presented in the following table:

M_0	T_0 (°F)	$S_{f_{min}}$	C_T
1.81	54	3.75	0.430
2.00	-79	2.72	.570
2.00	14	2.95	.525
2.00	48	3.26	.475

In the flight tests of this engine reported in reference 2, the over-all value of specific impulse of 1059 pounds of thrust per pound of fuel per second was obtained for the powered portion of the flight. The conversion of this value of specific impulse to S_f gives a value of $S_f = 3.40$.

The air impulse efficiency of a ram-jet engine is defined as the ratio of the actual propulsive force at the nozzle exit to the calculated propulsive force for 100-percent heat release of the fuel. The air impulse efficiency η_i is presented in figure 9 as a function of fuel-air ratio. This parameter is useful in evaluating the performance of a ram-jet engine because all internal losses are accounted for in the determination of η_i. The calculation scheme for obtaining η_i is given in appendix C. A maximum value of η_i of approximately 93 percent was realized at fuel-air ratios near 0.03. The slight scatter present in the test points is within the accuracy of these tests and does not result from changes in free-stream static temperature. It will be noted that, over the range of fuel-air ratios for these tests, values of impulse efficiency only vary between 87 percent and 93 percent.

The combustion efficiencies η_c computed from test data are presented as a function of fuel-air ratio in figure 10. A peak combustion efficiency of 81 percent was obtained at a fuel-air ratio of 0.03. Flight tests of this engine (reference 2) showed an average value of η_c of 81 percent. The data show that any change in η_c due to change in free-stream temperature is small.

SUMMARY OF RESULTS

The important results obtained in free-jet tests of a 6.5-inch-diameter ram-jet engine at Mach numbers of 1.81 and 2.00 are summarized as follows:

1. A maximum thrust coefficient of 0.925 was obtained at a Mach number of 2.00 for a free-stream static temperature of -79° F at the fuel-air ratio of 0.07. The maximum thrust coefficient at a Mach number of 1.81 was 0.700 for a static temperature of 54° F and a fuel-air ratio of 0.100. The thrust coefficients were more than adequate to insure successful flight tests of the ram-jet missile at a Mach number of 2.00.

2. The lowest specific fuel consumption was 2.72 obtained at a Mach number of 2.00 at a free-stream static temperature of -79° F and a thrust coefficient of 0.570.

3. Maximum values of air impulse and combustion efficiency were 93 percent and 81 percent, respectively, and occurred near a fuel-air ratio of 0.03. Over the range of static temperatures tested, the combustion efficiency was not significantly affected by temperature.

4. Values of combustion efficiency and specific fuel consumption obtained in ground tests were of the same order of magnitude as those values obtained from the flight test.

5. At a Mach number of 2.00 and for a given thrust coefficient, excellent agreement in diffuser total-pressure recovery between flight- and ground-test data is noted.

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APPENDIX A

DETERMINATION OF TARE DRAG

In the drag tests, the force on the model measured by the thrust stand is made up of tare (external), internal, and base drag.

One-dimensional-flow theory gives the following expression for internal drag

$$D_i = \gamma p_0 M_0^2 A_0 + p_0 (A_6 - A_b) - p_6 (A_6 - A_b)(1 + \gamma M_6^2) \tag{1}$$

The value of p_6 can be found from the continuity equation to solve equation (1), with $M_6 = 1$ and A_0 calculated from appendix B.

$$p_6 = \frac{p_0 A_0 M_0 \sqrt{1 + \frac{\gamma - 1}{2} M_0^2}}{A_6 M_6 \sqrt{1 + \frac{\gamma - 1}{2} M_6^2}}$$

The measured base pressure p_b was used to calculate the base drag

$$D_b = A_b (p_0 - p_b) \tag{2}$$

The tare drag is obtained by subtracting the values of internal drag (equation (1)) and base drag (equation (2)) from the total force measured on the thrust stand.

APPENDIX B

STREAM-TUBE AREA RELATIONSHIP A_0/A_3

FOR CALCULATING AIR MASS FLOW

The external shock configuration in front of the inlet of this ram-jet engine is shown in figure 2(d). When this configuration (which is caused by the internal contraction of the inlet) is established, an expression relating A_0 and A_3 to air mass flow can be determined. The relationship between total pressure and critical area from the continuity equation for adiabatic flow is

$$H_0A_0^* = H_3A_3^*$$

Multiplying both sides of this equation by A_0, A_3, and H_1 and transposing gives

$$\frac{A_0}{A_3} = \frac{A_0}{A_0^*}\frac{A_3^*}{A_3}\frac{H_1}{H_0}\frac{H_3}{H_1} \tag{3}$$

To solve for $\frac{A_0}{A_3}$, the following ratios obtained from reference 4 must be used

$$\frac{A_0}{A_0^*} = \frac{(5 + M_0^2)^3}{216M_0}$$

$$\frac{H_1}{H_0} = \left(\frac{6M_0^2\sin^2\theta}{M_0^2\sin^2\theta + 5}\right)^{3.5}\left(\frac{7M_0^2\sin^2\theta - 1}{6}\right)^{-2.5}$$

And, if a negligible loss in total pressure between stations 2 and 3 is assumed,

$$\frac{H_2}{H_1} = \frac{H_3}{H_1}$$

or

$$\frac{H_2}{H_1} = \left(\frac{6M_1{}^2}{M_1{}^2 + 5}\right)^{3.5}\left(\frac{7M_1{}^2 - 1}{6}\right)^{-2.5}$$

where

$$M_1{}^2 = \left(\frac{M_{1_c} + M_{1_s}}{2}\right)^2$$

At station 3 the velocity is assumed sonic and the value of M_3 is 1. Hence,

$$\frac{A_3}{A_3{}^*} = 1$$

The experimental determination of A_0/A_3 can be obtained using data from the tare tests and the continuity equation

$$\frac{A_0}{A_3} = \frac{0.98H_6 \frac{p_6}{H_6}(A_6 - A_b)M_6\sqrt{1 + \left(\frac{\gamma - 1}{2}\right)M_6{}^2}}{A_3H_0 \frac{p_0}{H_0} M_0\sqrt{1 + \frac{\gamma - 1}{2} M_0{}^2}}$$

with the constant of 0.98 from reference 5 with the exit plug installed.

The energy equation gives

$$\frac{p}{H} = \frac{1}{\left(1 + \frac{\gamma - 1}{2} M^2\right)^{3.5}}$$

Substituting gives

$$\frac{A_0}{A_3} = \frac{0.98H_6\left(A_6 - A_b\right)\dfrac{M_6}{\left(1 + \frac{\gamma - 1}{2} M_6{}^2\right)^3}}{A_3H_0\dfrac{M_0}{\left(1 + \frac{\gamma - 1}{2} M_6{}^2\right)^3}} \tag{4}$$

If isentropic flow is assumed to exist between stations 4 and 6, then

$$H_4 = H_6$$

and the integrated average of H_4 weighted against area is substituted in equation (4).

Values of A_0/A_3 calculated from equation (3) are presented in figure 11 for several free-stream Mach numbers. Experimental verification of this method obtained from drag-run data at $M = 1.81$ and 2.00 are included in figure 11. Good agreement between calculated and experimental values of A_0/A_3 is noted.

APPENDIX C

DETERMINATION OF RAM-JET PERFORMANCE PARAMETERS

The air specific impulse is defined as

$$S_a = \frac{\rho A V^2 + pA}{W_a}$$

$$= \frac{g\rho A V}{W_a}\left(\frac{V}{g} + \frac{p}{\rho V g}\right)$$

As $W_a + W_f = \rho A V g$, then

$$S_a = \left(1 + \frac{f}{a}\right)\left(\frac{V}{g} + \frac{p}{\rho V g}\right)$$

At the throat (station 5) $M_5 = 1$. Then

$$V_5 = \sqrt{\gamma_5 g R_5 T_5}$$

and

$$\frac{p_5}{g\rho_5} = R_5 T_5$$

Substitution gives an expression for the air specific impulse

$$S_{a_5} = \left(1 + \frac{f}{a}\right)\left(\sqrt{\frac{\gamma_5 R_5 T_5}{g}} + \sqrt{\frac{R_5 T_5}{\gamma_5 g}}\right)$$

$$= \left(1 + \frac{f}{a}\right)\sqrt{\frac{R_5 T_5}{\gamma_5 g}\left(\gamma_5 + 1\right)^2} \tag{5}$$

Values of T_5 and γ_5 for various fuel-air ratios and initial air temperatures have been calculated by using reference 6. Equation (5) provides means of calculating the impulse at the throat of the exit nozzle.

The effect of γ varying from stations 5 to 6 is negligible and γ is therefore assumed to be constant in order to simplify computation. This leads to an equation obtained from reference 7 which gives a factor by which the impulse at the throat is increased through the expanding section of the nozzle.

$$\phi M = \frac{1 + \gamma_5 M_6^2}{M_6} \frac{1}{\left[2\left(1 + \gamma_5\right)\left(1 + \frac{\gamma_5 - 1}{2} M_6^2\right)\right]^{1/2}} \tag{6}$$

where M_6 is found from the continuity relationship

$$\frac{A_6}{A_5} = \frac{1}{M_6}\left(\frac{1 + \frac{\gamma_5 - 1}{2} M_6^2}{1 + \frac{\gamma_5 - 1}{2}}\right)^{\frac{\gamma_5 + 1}{2(\gamma_5 - 1)}}$$

The exit jet propulsive force is

$$G_6 = S_{a_5} \phi M W_a \tag{7}$$

The impulse efficiency is a ratio of the actual propulsive force at the exit of the nozzle to the calculated propulsive force for 100-percent heat release of the fuel.

The thrust of the ram-jet engine is given by

$$F = X + D_t = G_6 - p_0\left(\gamma_0 M_0^2 A_0 + A_6\right)$$

Then, the actual value of exit jet propulsive force is

$$G_6 = X + D_t + \gamma_0 p_0 M_0^2 A_0 + p_0 A_6 \tag{8}$$

The ideal exit jet propulsive force is

$$G_6' = \phi M S_{a_5}' W_a$$

Then

$$\eta_i = \frac{G_6}{G_6'} \tag{9}$$

To compute the combustion efficiency, the value of air specific impulse at the throat S_{a_5} is found from equations (3), (6), (7), and (8). From the values of S_{a_5}, the stagnation temperatures at station 5 may be determined by use of equation (5) and data calculated by using reference 6.

Reference 8 provides a means of calculating the heat release per pound of ethylene Δh as a function of the temperature rise. Then

$$\eta_c = \frac{\Delta h}{\frac{F}{A} h_c} \tag{10}$$

where h_c is taken as 20,400 Btu per pound for ethylene.

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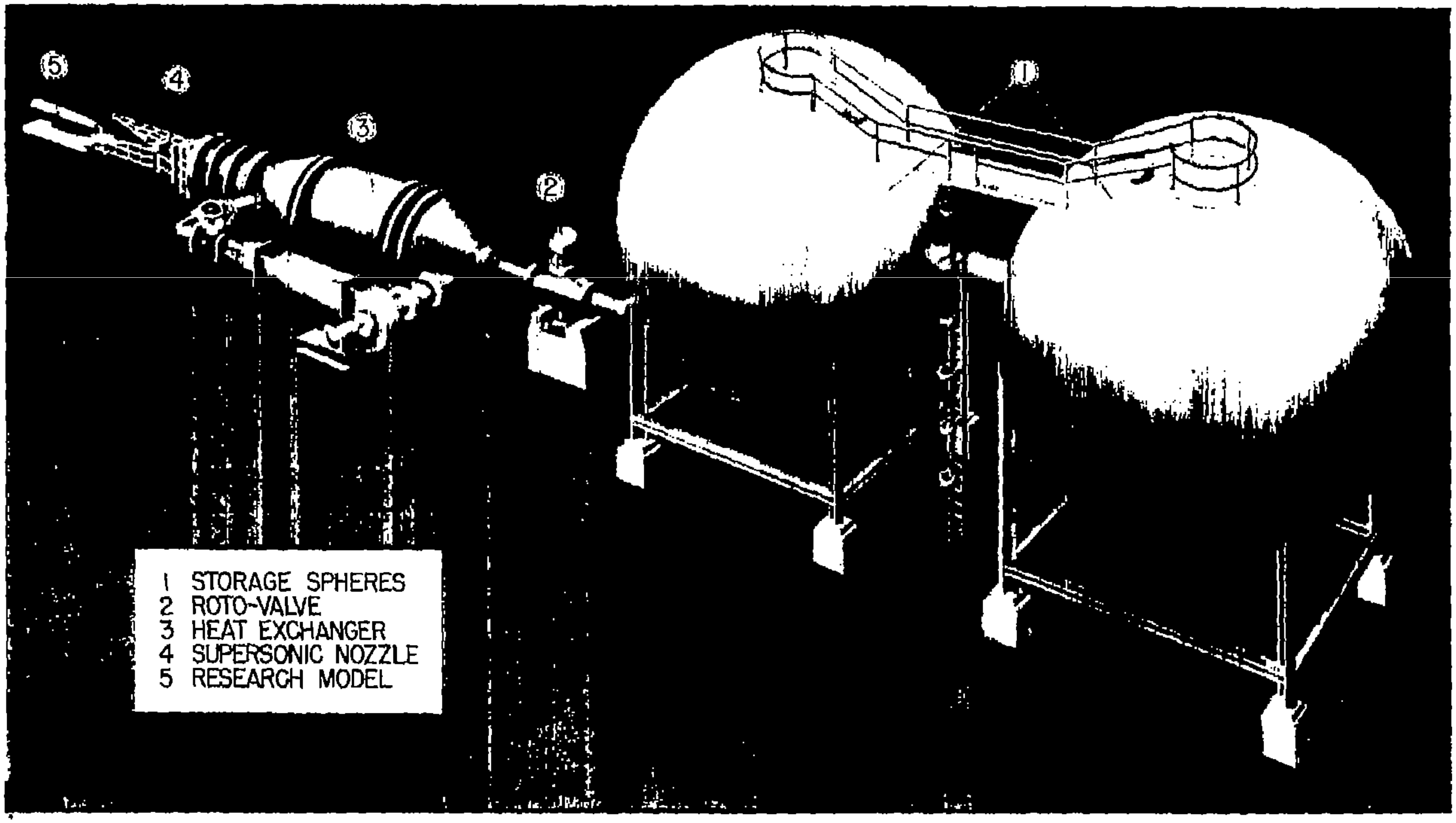

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Figure 1.- The preflight jet facility at the Pilotless Aircraft Research Station.

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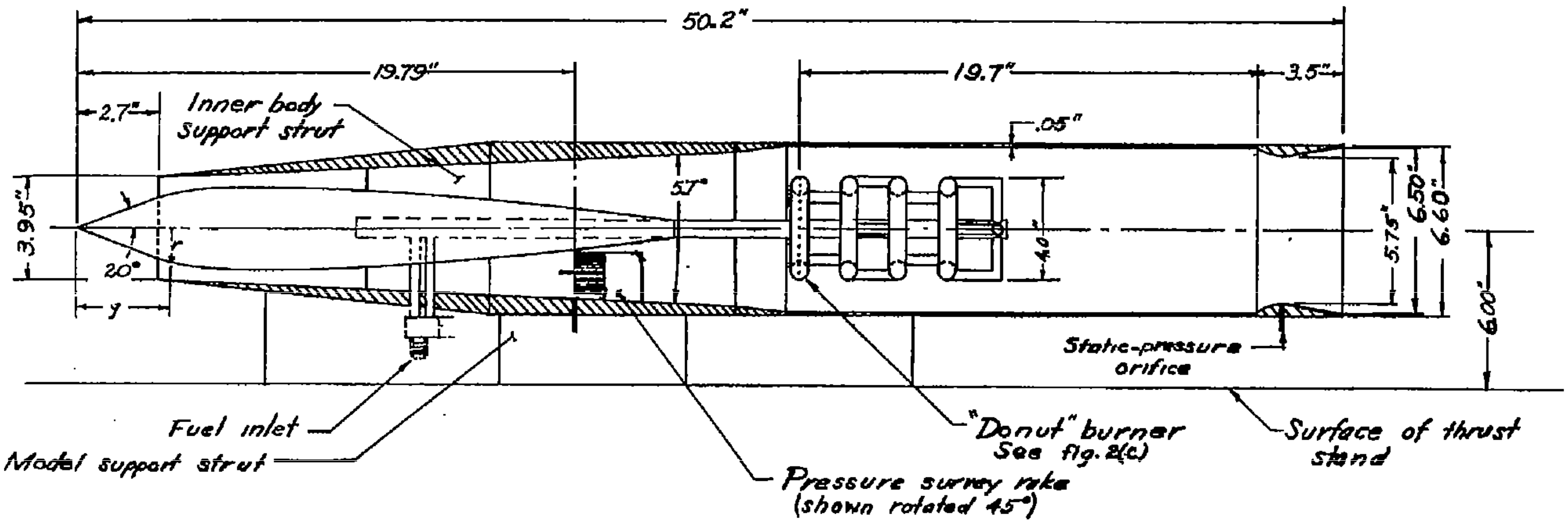

Inner Body Ordinates														
Y, in.	0	3.300	3.400	3.500	3.600	3.700	4.200	4.700	5.200	5.700	6.700	7.700	8.700	9.700	10.700
r, in.	0	1.200	1.232	1.254	1.269	1.280	1.320	1.350	1.370	1.380	1.380	1.375	1.370	1.360	1.345
Y, in.	11.700	12.700	13.700	14.700	16.000	17.000	18.000	19.000	20.000	21.000	22.000	23.000			
r, in.	1.325	1.300	1.270	1.230	1.170	1.100	1.020	0.930	0.830	0.720	0.600	0.470			

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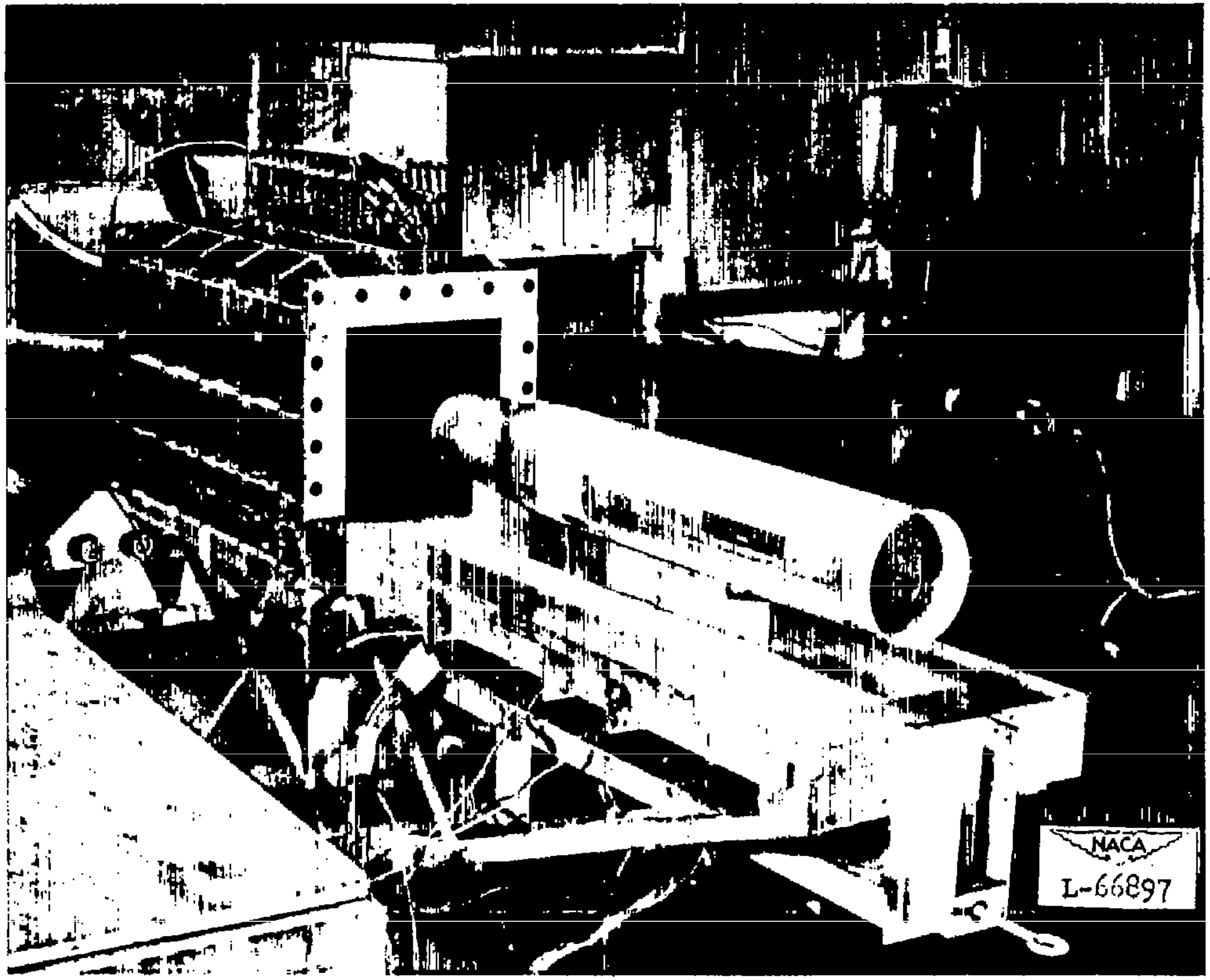
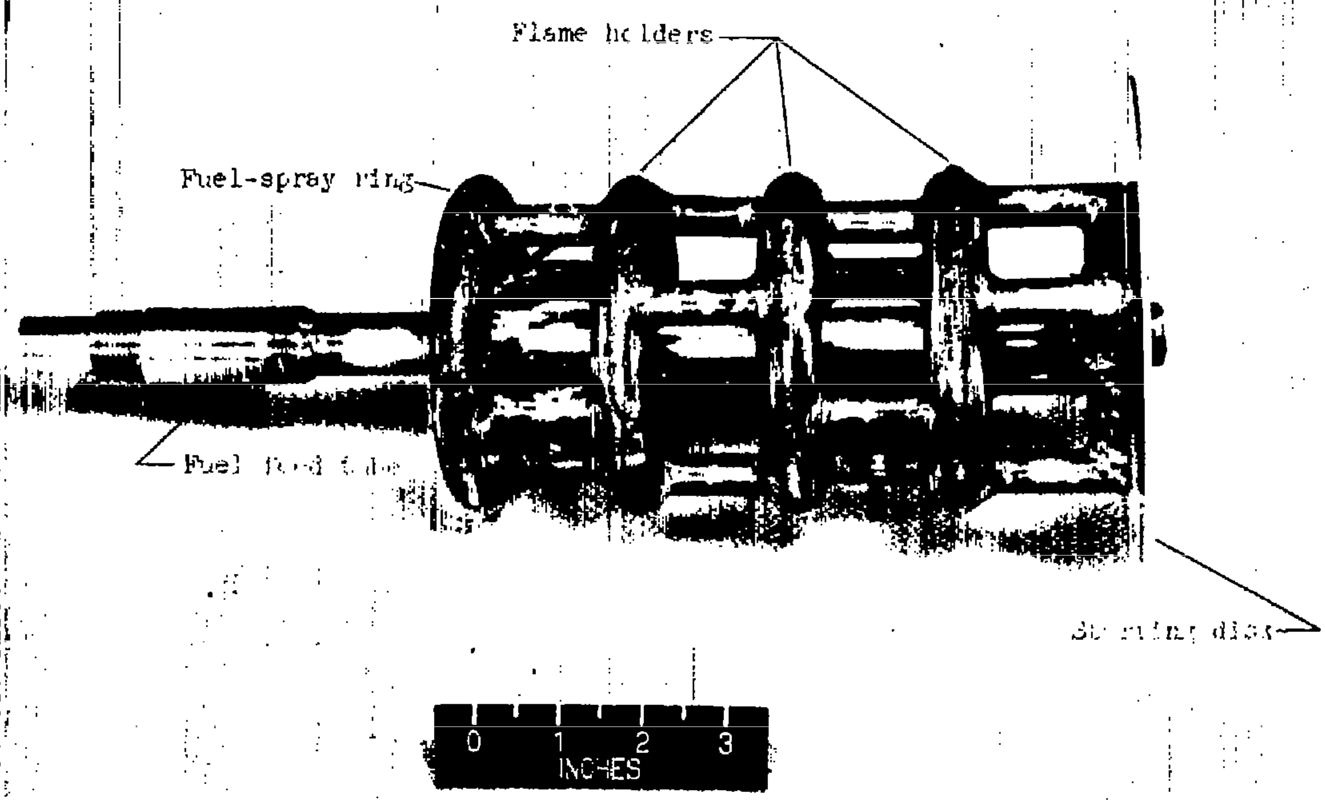
(a) Arrangement of ram-jet model showing principal dimensions and pressure-measuring stations.

Figure 2.- The ram-jet engine used for performance tests.



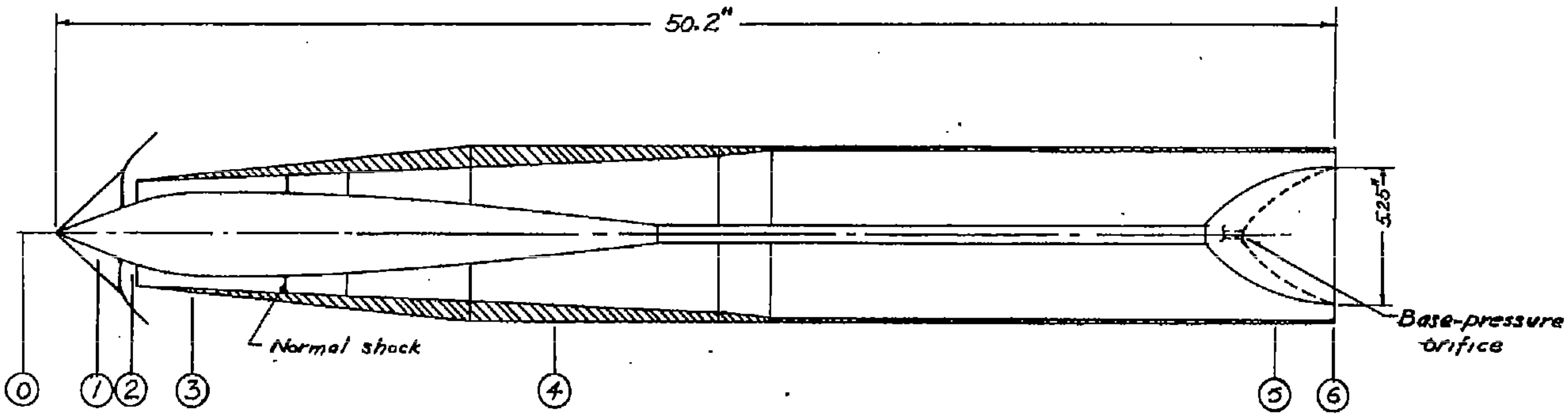
(b) The ram-jet engine mounted in the preflight jet.

Figure 2.- Continued.



(c) Three-row donut burner with starting disk.

Figure 2.- Continued.



Stations where pressure was measured
and data were analyzed

0	Free stream
1	Behind oblique shock
2	Entrance
3	Minimum area
4	Rake
5	Nozzle throat; performance tests
6	Exit

(d) Tail-plug installation for drag tests and locations of stations used in data analysis.

Figure 2.- Concluded.

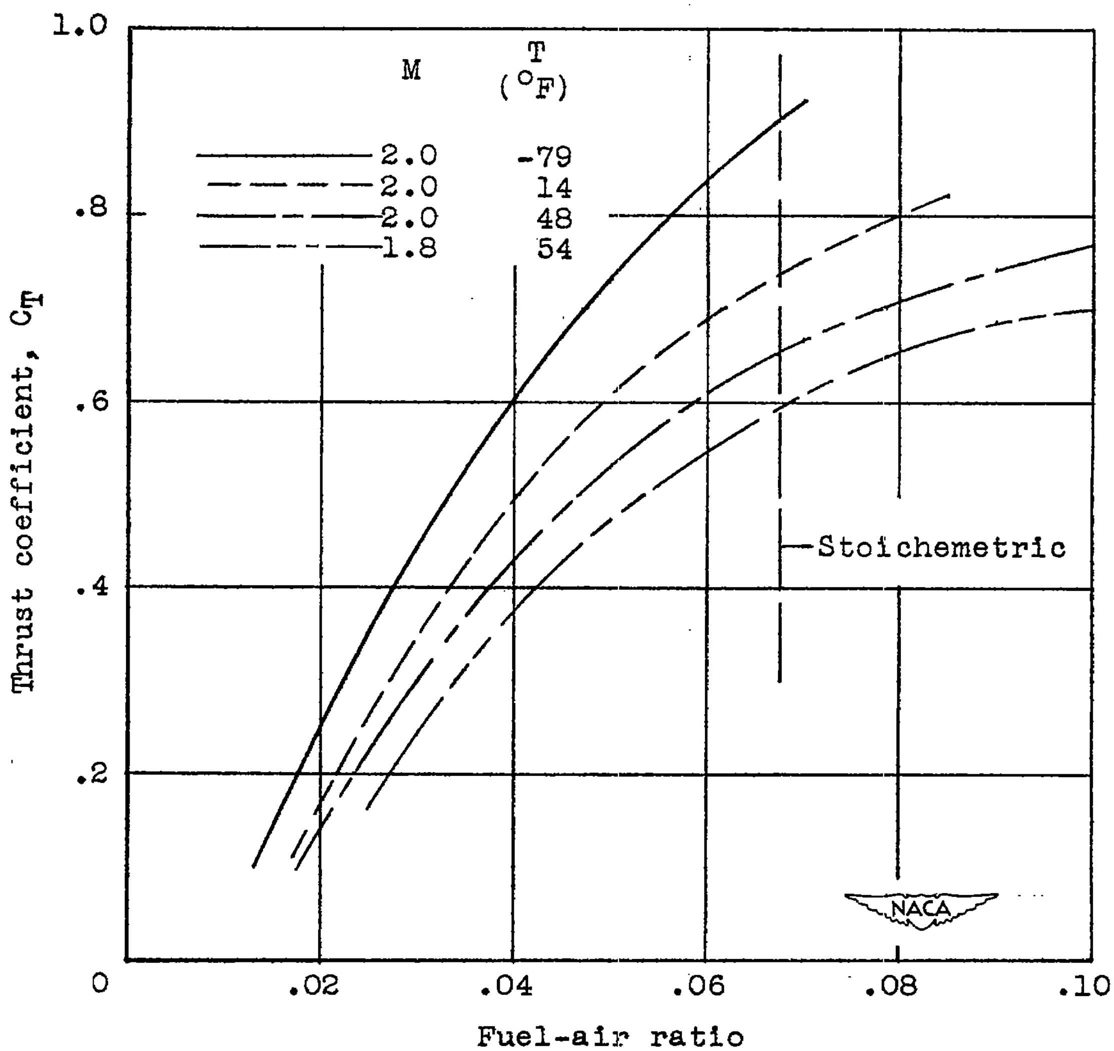


Figure 3.- Thrust characteristics of the ram-jet engine at $M = 1.81$ and 2.00 at several values of free-stream static temperature.

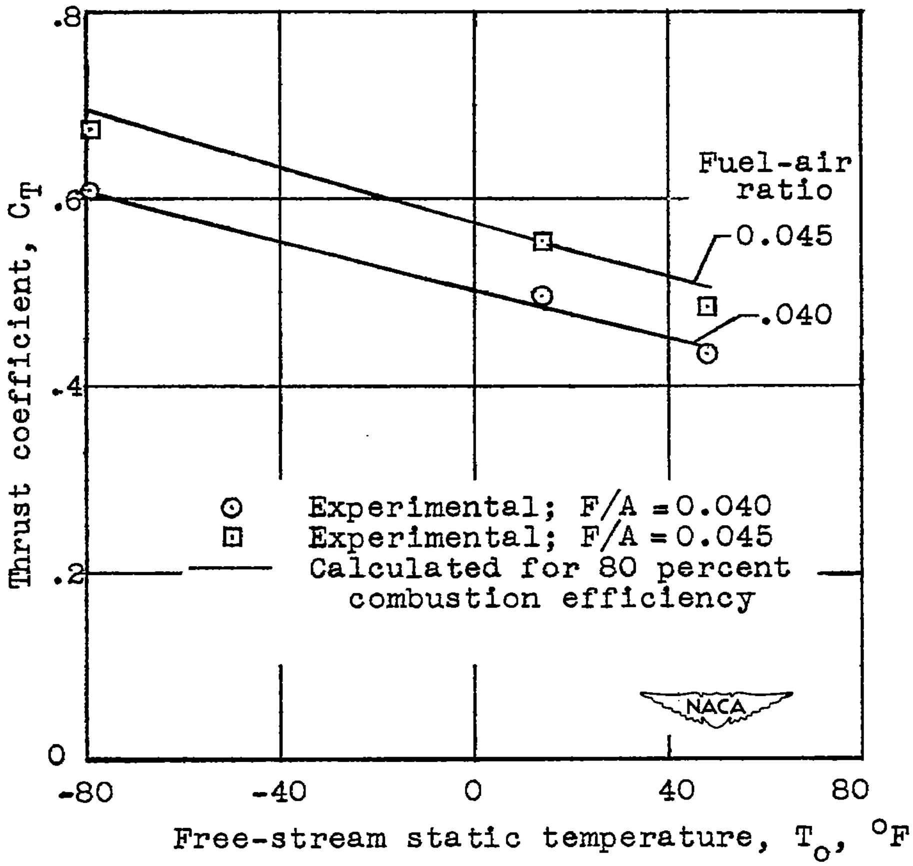


Figure 4.- A comparison of experimental and calculated thrust coefficient C_T for various values of free-stream temperature. M = 2.00.

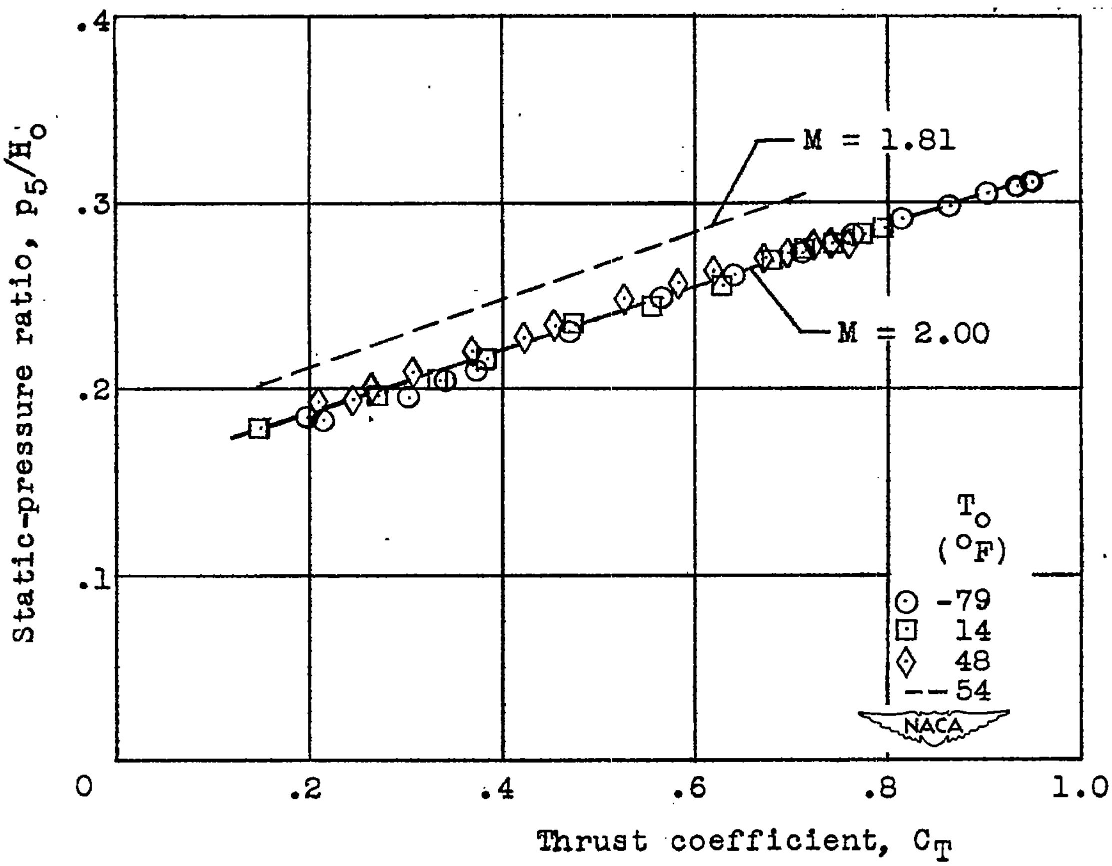


Figure 5.- Nozzle-throat static-pressure coefficients at M = 1.81 and 2.00 for different values of free-stream static temperature.

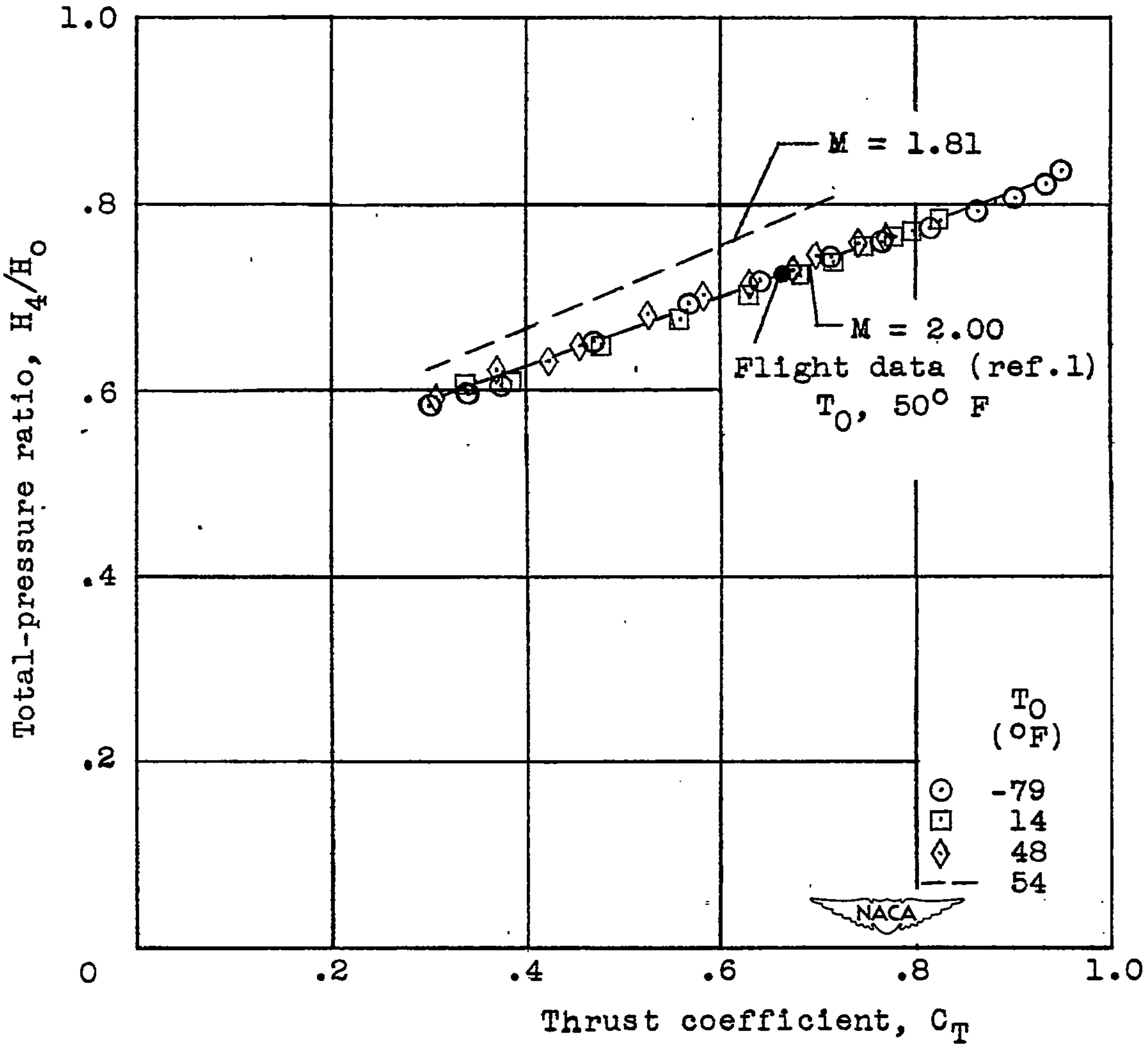


Figure 6.- Diffuser total-pressure-recovery ratios obtained at M = 1.81 and 2.00 at several values of free-stream static temperature.

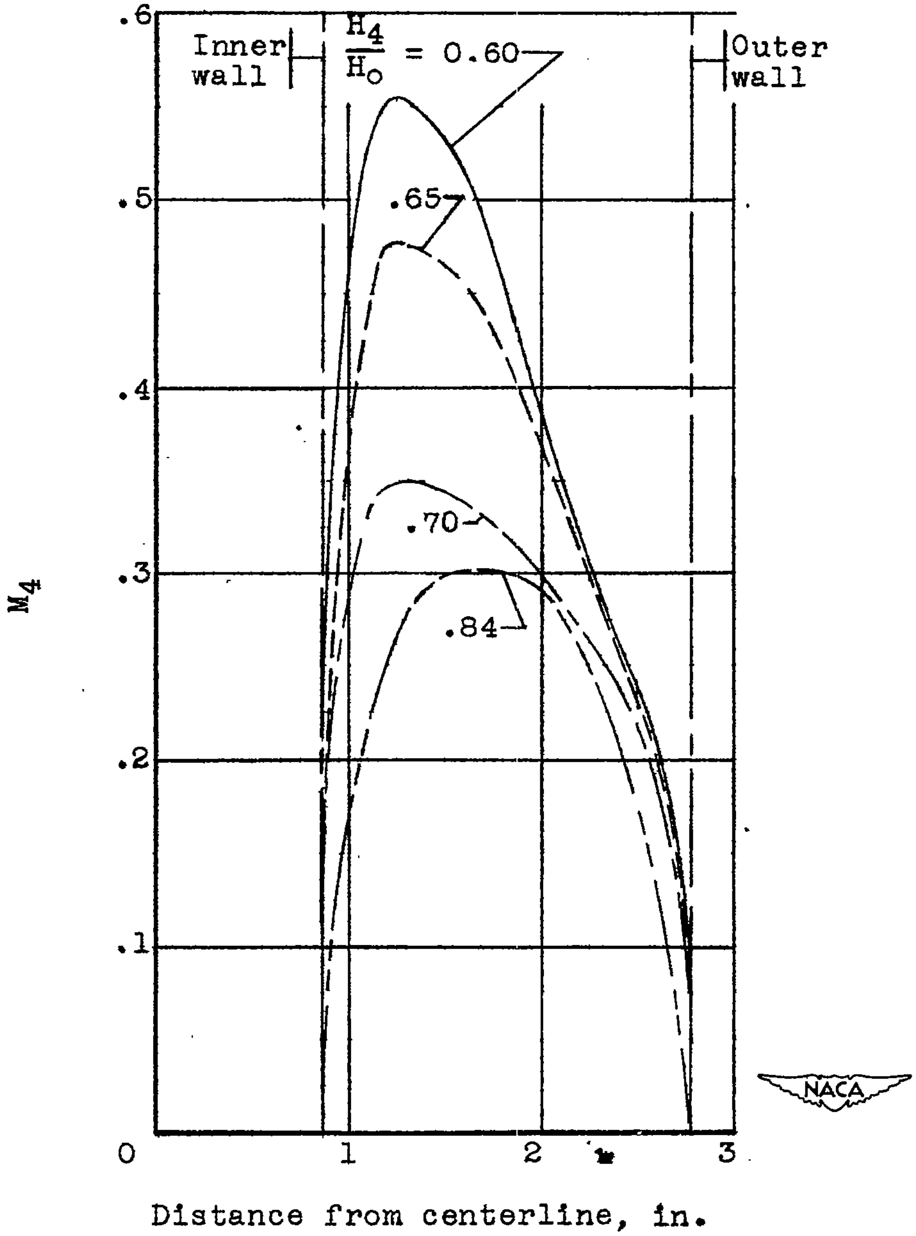


Figure 7.- The radial distribution of Mach number at station 4 for several values of H_4/H_o.

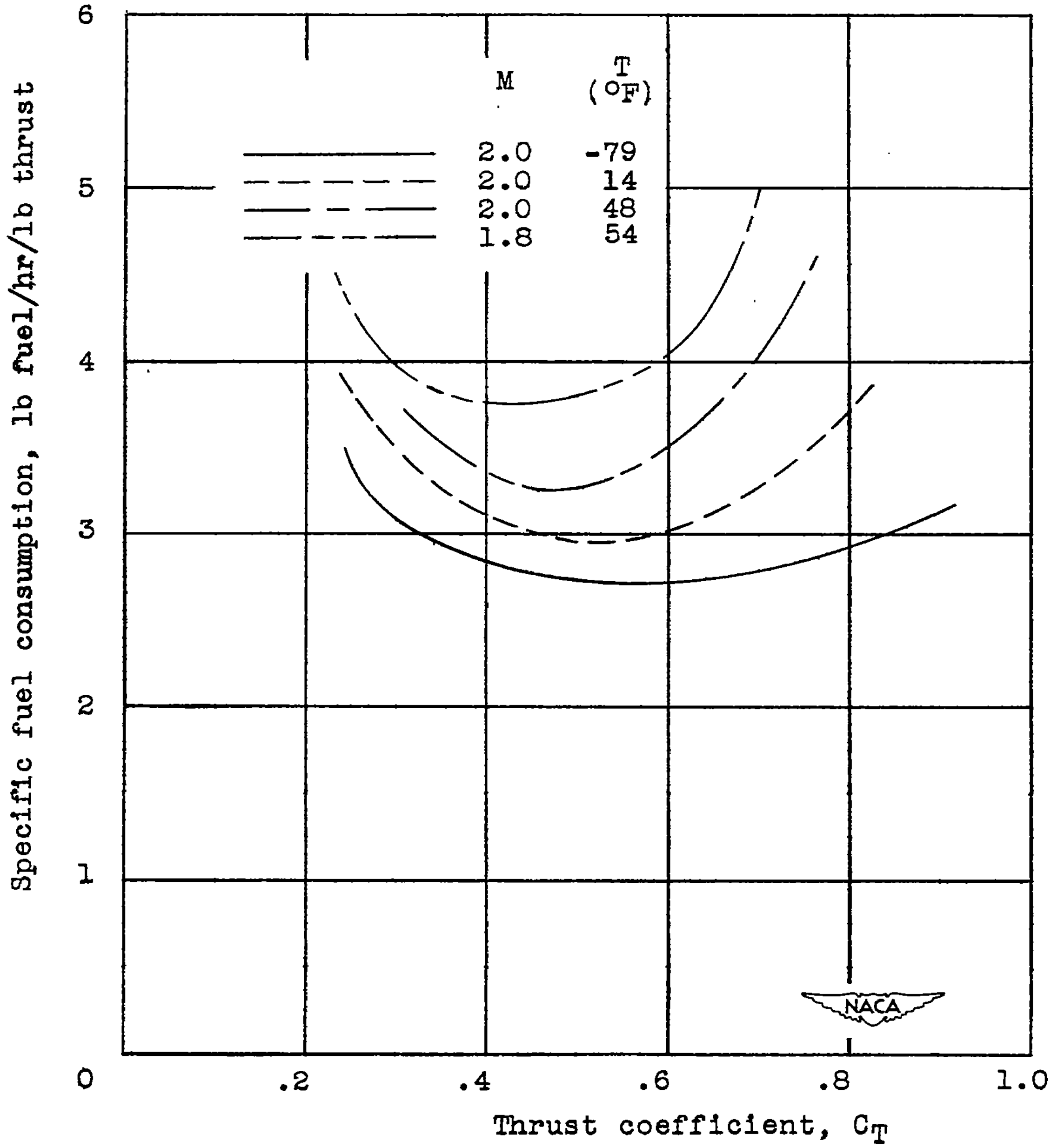


Figure 8.- The economy of the ram-jet engine as a function of thrust coefficient obtained at M = 1.81 and 2.00 for several values of free-stream static temperature.

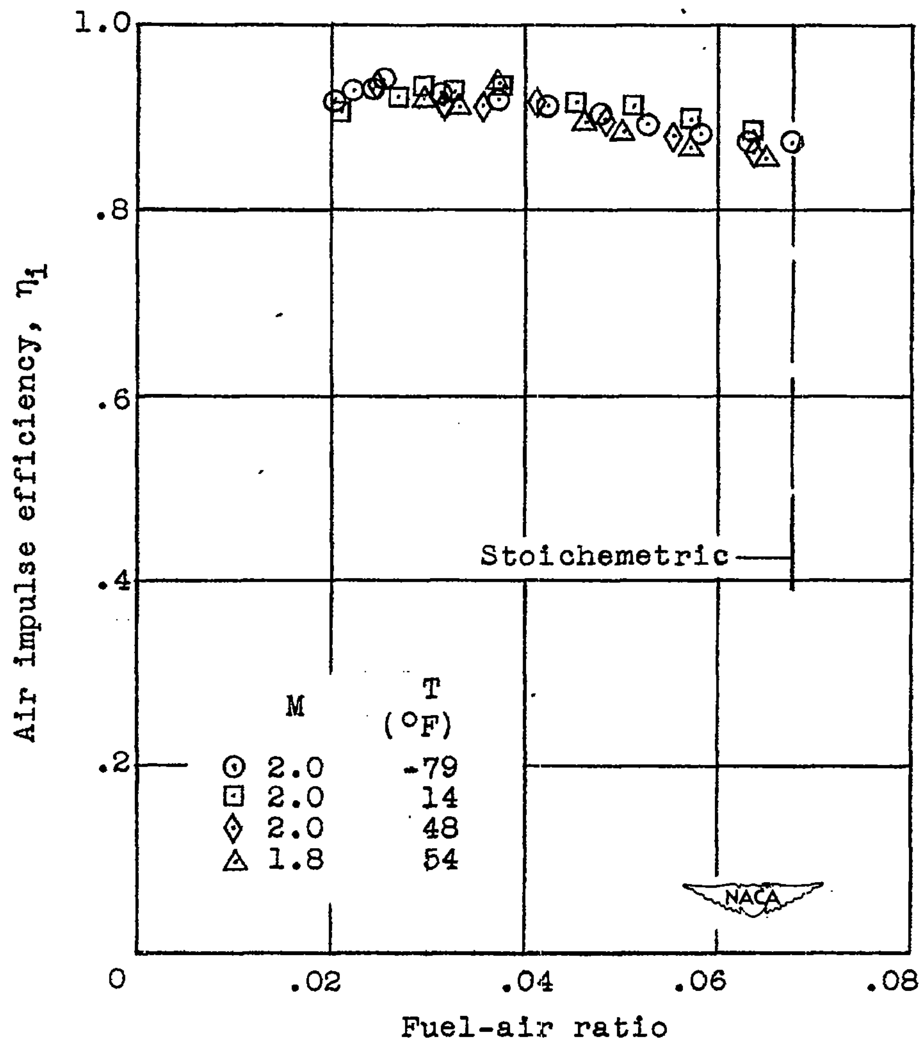


Figure 9.- Air impulse efficiencies obtained in tests of the ram-jet engine at M = 1.81 and 2.00 for several values of free-stream static temperature.

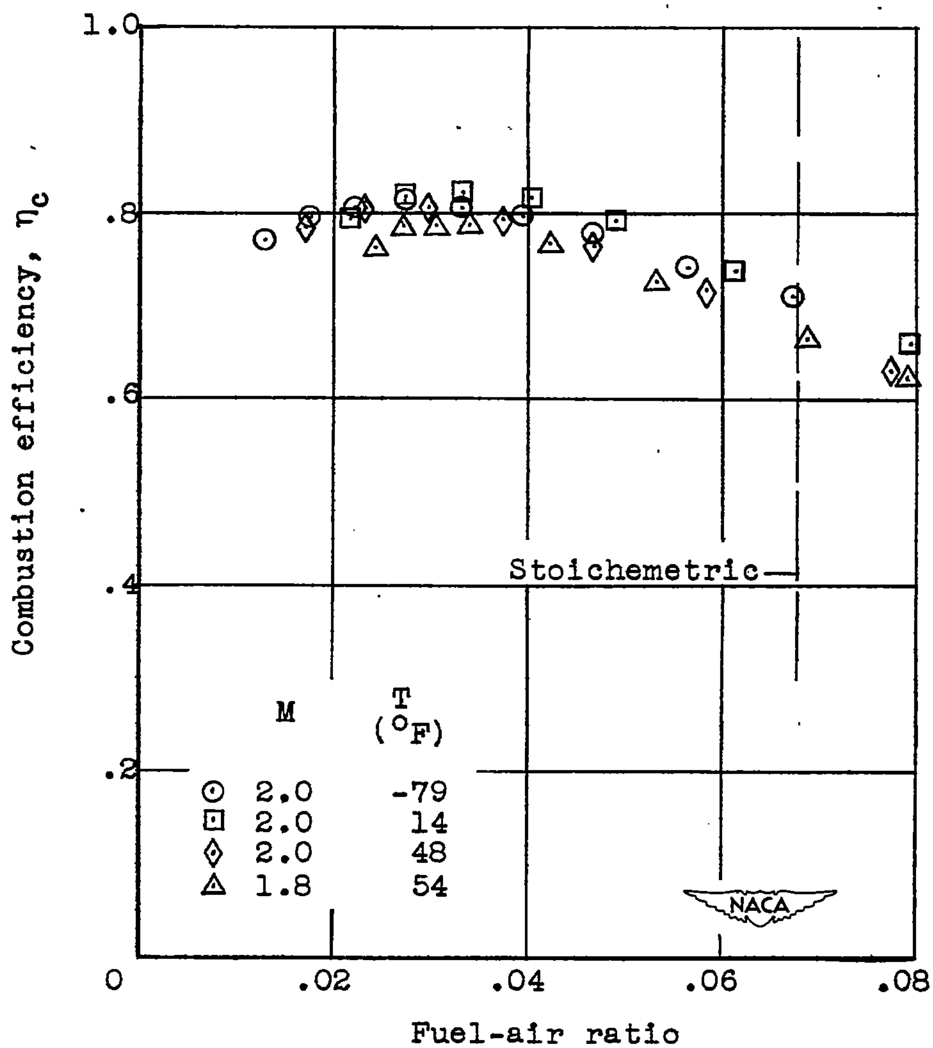


Figure 10.- The relationship of combustion efficiency and fuel-air ratio for the ram-jet engine at M = 1.81 and 2.00 at several values of free-stream static temperature.

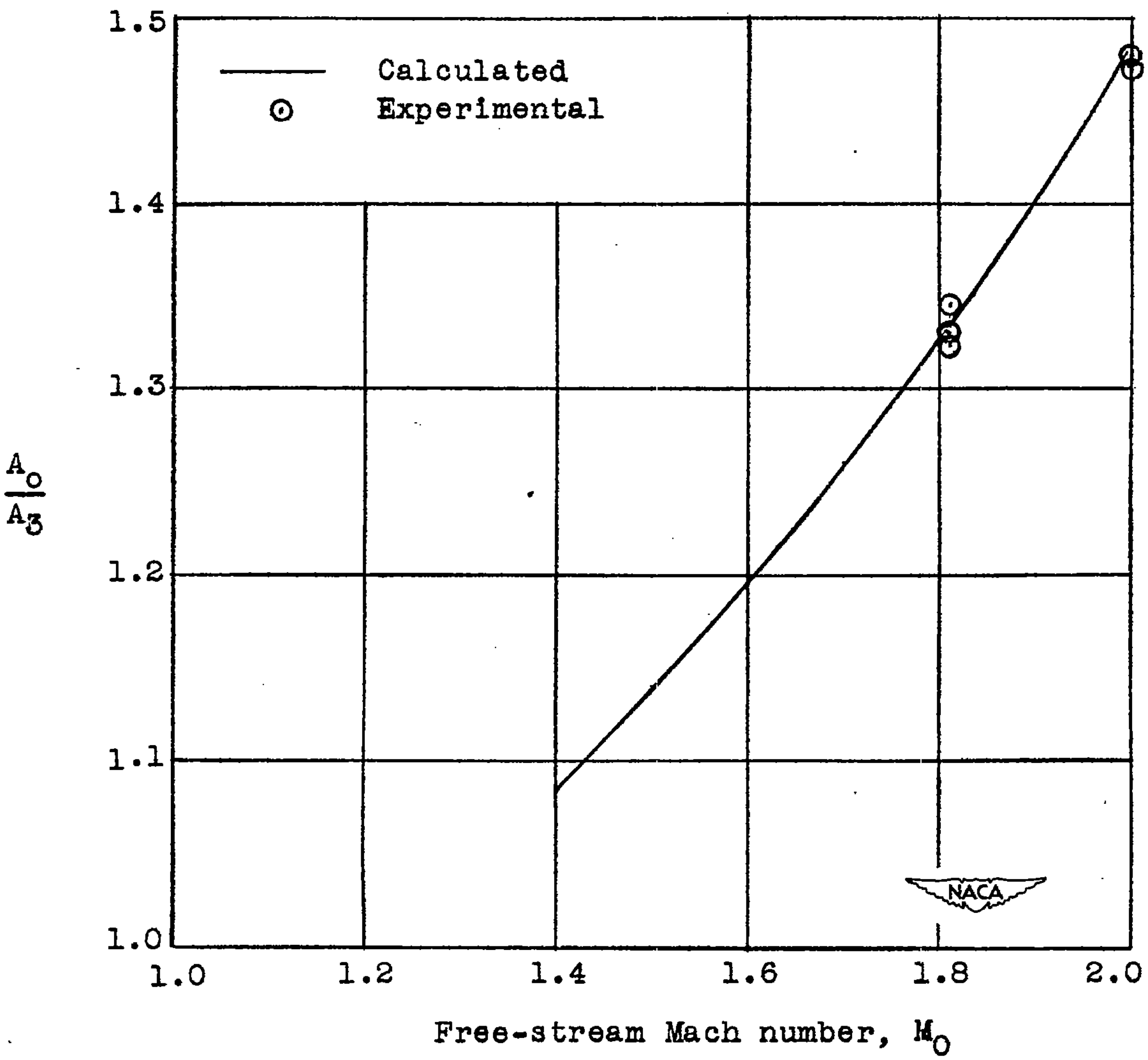


Figure 11.- Air mass-flow parameter as a function of free-stream Mach number.

NACA-Langley - 3-7-51 - 550